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ROBERTSON YOUNGQUIST, Editor

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PROBLEMS FACING THE ROCKET INDUSTRY RELATING TO MILITARY PLANNING

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BEFORE beginning this discussion, it may be well to spend a second limiting the field to be discussed. Terminology in the rocket field is so broad that unless definitions are given, speaker and listener are often not on common ground. Basically there are two types of rocket engines—those using solid fuels and those using liquid fuels. The phrase "rocket engine" as used throughout this paper refers only to the liquid-propellant rocket engine.

The liquid-propellant rocket-engine industry is a baby industry, both from the standpoints of relative age and size. Ten years ago there was no company in the United States engaged in the development and fabrication of liquid-propellant rocket engines. Today, although there are some 15 companies in this country evidencing an interest in rocket development, many of these companies did not begin any work in this field until the latter stages of World War II. Even though some of the more recent entries into the rocket field are substantial members of the aeronautical field in other product lines, it may be conservatively estimated that less than 4000 people make up industry's total rocket-engine effort today.

The rocket engine is part of the jet-engine group, which covers a broad field of power sources for air-borne vehicles and no part of which antedates the formation of the rocket-engine industry to any extent. During the same period of early growth of the jet-engine industry, the acceptance of the guided missile as a military weapon took place. Jet-engine developments and guided-missile developments are so interlocked that each has a considerable effect on the other, and it is difficult to speak of one without referring occasionally to the other. A great amount of effort has been expended in these two fields in recent years, not only because of the potential applications of new types of power and new types of weapons, but also to catch up with developments in these fields, where, frankly, we were considerably behind other countries. If we are to believe the rumblings from across the sea which the newspapers have carried recently in regard to the Russians being further along in the development of a long-range missile than we are,

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the immediate future of the jet-engine and guided-missile industry, and in particular the rocket-engine industry, may continue to be on a "hurry-up" basis.

It is always a sound idea to step aside and take stock of where one has been and is going. It is even more necessary that such a review and consideration be given to an industry that has been, and apparently will continue to be, on a "push" program basis. A complete, detailed review of the rocket-engine industry would run over the time allotted for this presentation, so this paper will be confined to those problems facing industry in the development of the rocket engine, which, in my opinion, are the most pressing.

All of these problems deal with the military aspect of the rocket engine, for today's utilization of rockets is almost entirely in this connection. The answers to these problems have a great bearing on military planning as applied to rocket engines—not from a requirements point, for that is not industry's business, but from a supply standpoint, which is definitely industry's role.

This discussion may be divided roughly into three broad groups: Problems of development, problems of production, and problems of economics.

Problems of Development

Power plants for aircraft and guided missiles are one of the most involved engineering problems in existence because of the high power requirements per lb of material, the severe thermal and mechanical stresses involved, the ability to operate over extreme altitude and temperature conditions, the necessity for relatively simple controls, and the reliability requirements. The rocket engine as a power plant faces all of these problems, compounded a bit by more extremes in everything mentioned except, perhaps, controls and reliability. Despite this similarity of the broad engineering problems, the development of rocket engines has not tended to follow the development course of the reciprocating and turbojet engines in the important field of standardization.

In the reciprocating and turbojet-engine industries, it is the customary practice to develop a given type of engine in varying power sizes which are then used as basic standards for application. Potential users, aircraft and missile designers, normally utilize engines from such an existing group. Admittedly, under these circumstances, some compromises are made in design of the end-use article in weight, performance, or range. In the rocket-engine industry, however, it has generally been the practice that each liquid-propellant rocket engine developed must incorporate the highest available performance with the lowest possible weight for its first application. As a result, the rocket engine ends up as a specific design for a specific application, and is frequently not usable in its normal basic form for any other application. Aside from the loss of savings possible through stand-

ardization of any sort, this practice has tended to place an unusually heavy development burden on the rocket-engine industry, in that every engine is new, practically down to the last nut and bolt.

It is understandable that this condition is the result of rapid development growth both in the rocket-engine field and in the rocket-powered airplane or missile field. The variations in development of rocket engines due to utilization of the various propellants which are considered advantageous from either an efficiency or logistics standpoint are sufficient in themselves to make real standardization difficult today. Furthermore, standardization of power-plant sizes is practical only when the immediate future end-use article development requirements can be reasonably forecast; today, a very difficult task.

While no major steps toward standardization of types and sizes of rocket engines may be undertaken at this point for the reasons stated, it is extremely important that we do not become so accustomed to the "special" design concept that steps along that road are not taken as soon as possible. Recently there has been consideration of attempting to standardize somewhat through the medium of the rocket-engine thrust chamber, with the idea in mind that rocket engines will be made up of various numbers and sizes of standard thrust chambers. While this definitely is a move in the right direction, it still leaves a long way to go. Changing the number of thrust chambers on a developed rocket engine is sometimes, on a relative basis, analogous to changing the number of cylinders on a reciprocating engine.

The drive toward standardization of some types will probably be impeded by the fact that the development cost of a rocket engine to the point of securing an operating prototype, while not inexpensive, is considerably less than similar costs for other engines except, perhaps, the pulsejet and the ramjet. This relative economy of engine cost, together with the aircraft designer's normal impulse to secure the last shred of performance from the final product, the airplane or missile, will make it very tempting to continue the special single-purpose rocket-engine design. However, the development cost of a rocket engine is not so inexpensive that it should always be the variable in an end-product design.

Engine-size standardization, which should be the first step, is under consideration in part. To make the determination of "standard" sizes, even on an interim basis, is not easy; yet, not to make the decision may be far more costly than is realized. Standardization of engine sizes would not tend to throttle development, which for a time must keep on at its rapid pace. The time to take this first step in standardization would appear to be here.

Standardization of Propellants

Standardization of rocket-engine propellants will most likely continue to

be deferred for a time, because it is in this field that the greatest advances in engine performance are possible. In this connection it is disturbing to note that the logistics of supply of the propellants as the supply now exists apparently has great weight in determining the course of the development of rocket engines. While the logistics of supply cannot be ignored, may it not be that development or expansion of propellant supply to fit existing rocket-engine requirements is as easy and economical as the converse?

The first two phases of the development of a rocket engine, the prototyping and the experimental production phases, are being carried out on a critically short-time cycle. This situation exists in part because of the demands for rocket engines incorporating the latest basic advances in the art, particularly those along the improved performance line. It will be necessary for the industry to live with this problem in so far as it is caused by new basic developments, because the need for the Armed Services to maintain a position of leadership in aircraft and missiles makes it necessary that applications of new research data be applied while the research investigation is still in process. This short-time cycle, however, is in part due to contract limitations. Because of the small number of rocket engines of any one type produced, the impracticality of predicting future quantity requirements on a reasonably certain basis, and the unusually high development risks involved, most rocket-engine development work has been, and will probably continue to be, undertaken under government contracts. These development contracts use appropriations which usually have a life of two years, plus the year of appropriation, which determines the time cycle. The time element for the development of an experimental airplane is from two to four years. The time element for the development of a prototype reciprocating or turbojet engine is approximately three years. The rocket-engine development, operating with much less basic research data than available to either of these two fields, can hardly be expected to be accomplished in less time. Peenemunde, the German Government rocket development and testing base, was reported in operation in 1937. By 1939 the preliminary plans for the V-2 were laid and by 1941 the first experimental unit V-2 was tested. In 1944 the V-2 was finally put into operation. Under the stimulus of wartime conditions, it took two years from plans to test of the first experimental unit, and three more years before the unit was placed in operation. It is rare that such schedule durations have been found in our peacetime government contracts because of appropriation expiration. I do not consider that only to equal the "other fellow's" performance is a good criterion. Just because the Germans took five years is no reason in itself that a similar job could not be done in less time by others. Yet, I feel sure that the staff of Peenemunde did not take advantage of extended appropriations to be slow in their work.

Even though appropriations have been extended at times, the extension has come at such a late date that speed-up operations in the development

have already taken place. Development tends to seek a pace commensurate with the basic research data available. It can lag, but it cannot be effectively speeded up beyond its normal pace without taking short-cuts, which is generally an uneconomic procedure and has to be paid for later. Great strides in improving the contract-time situation with respect to life of appropriations have been made, and it is hoped that the industry will soon see the results.

Even though the industry is going to be faced with the necessity of short development cycles from a military-requirements standpoint, some consideration of better cycle programming must be given. It has not been unusual that so-called production units, which in reality are experimental units with a fairly high degree of interchangeability, are required to be delivered within three to four months after the first successful operation of the prototype unit. The concentration of engineering effort in securing an operable experimental prototype unit incorporating the application of new research data does not make it possible that sufficient effort be put on the producibility of the design at the same time. Such a short-time cycle between first operating unit and delivery of experimental production units does not allow for much redesign. As a result, development at an above-normal level is continued into the experimental production phase, which not only affects deliveries but also over-all costs of the engine and the aircraft or missile using the engine. It is questionable whether any real gain is achieved under such programming.

Value of Field Experience

Perhaps the most important phase in the development of any type power plant is the actual day-in-and-day-out service use of the power plant in the field under actual operation conditions, backed by further intensive development work on problems shown to exist with such engines as a result of the field experience. The design of a power plant resulting after such field-service trials represents the really useful production power-plant design. The rocket engine of today's design suffers from a distinct lack of field operational experience except in a very few cases. It is frequently necessary for contractors to be well along in the development of a new and more complex engine before its predecessor type has had a chance for much field operation. Needless to say, operations of this type do not make it easy to eliminate design items which may cause difficulty in the field in subsequent designs.

This condition basically has been brought about by a restriction of funds for the procurement and operational testing of missiles and aircraft powered by liquid-rocket engines. If it were necessary to make a choice of absolute values, it would be difficult to say which is the wiser course in the expenditure of development funds: to keep reaching for higher and higher degrees of development through the route of new items, or to develop an existing

item more thoroughly. It should not be necessary to choose; both programs are necessary for proper balance. New basic developments must be pushed if our military position is to be maintained. But our military position is also based on producibility of articles and trained personnel for their utilization. The follow-through on more field operation of rocket engines is mandatory if rocket-engine designs are to progress along sound lines, and if we are to have a sufficient nucleus of trained operating personnel.

I am reasonably certain that the extension of this recommendation to the guided-missile field will bring up the point that the guidance-systems development status makes it necessary to hold back on more field operation. This approach is somewhat like that of the gardener who cannot plow today because he has no seed. Can the nation afford to put off engine and airframe work until tomorrow because the guidance system is not entirely ready yet? I think not. Let the engine and airframe field operation and development go as far as they can—the guidance-systems development can catch up later.

It will be expensive to make more engines and missiles and to fire them. It is also expensive to rush into new developments before the preceding similar development has had a good evaluation. Yet the expense of continuing rocket-engine development further into the field operational phase would be an invaluable investment when this country requires rocket engines during an emergency period.

This same restriction of funds for field operations frequently places the industry in the position of not being able to perform the intensive development work on such problems as are shown up by field operations actually conducted. Available money is largely used for new procurement, not for continued production development of an existing engine. Contract provisions and time limitations preclude the performance of such work under the basic development contract.

From the very beginning of the rocket-engine industry, development has been undertaken on a narrow basic research foundation. The basic data developed by Dr. Robert H. Goddard and by the early version of the American Rocket Society made up the bulk of this country's assets in this field prior to the war. The securing and evaluation of German data after the war made a considerable addition to this field. Further additions to basic research data are being made daily by the rocket-engine industry, a number of universities, and the development laboratories and evaluation centers of the Armed Services.

Despite the results of the work being done by these groups, I think that rocket-engine development is being extended too far beyond its research and theory base; or, conversely, research and theory are lagging behind application. As I have said, some extension of development beyond firm research is necessary for our military security; the degree of extension is the discussable point. More basic research in combustion characteristics,

the extension of heat-transfer and radiation theory and characteristics into the operating ranges of today's rocket engines, and the design parameters of propellant injectors should be undertaken.

Empiricism in Design

The degree of empiricism used in a development is dependent on the soundness of the basic theory applicable to the development. Because basic theory has not been soundly established for all operations of a rocket engine, or because characteristics under fundamentally sound theory have not been determined for operating ranges in use today, the degree of empiricism in rocket engines is very high. This degree of empiricism should be reduced, but not eliminated, for some empiricism is a good forerunner for theory.

While a program for the reduction of the degree of empiricism in the rocket-engine industry will be costly, it can be expected to effect considerable savings in the development activities in a reasonably short time. The rocket-engine industry is not financially able to underwrite such a program, and even if it were, the probability of recovery of such investment under the limited rocket-engine production of current programs is highly marginal. Under these conditions government sponsorship of the program is essential. Some of the basic research, such as that in the field of measurements, will be applicable to other developments, both military and peaceful, from which the Government will benefit.

The major problems of production in the rocket-engine industry stem from the lack of production orders. Current requirements for rocket engines will not support production orders; small service-evaluation-type orders are ample. The rate of new engine developments and developments of engine applications is expected to continue at its present level, which will tend to minimize production requirements. Small quantities of new types of engines will probably be the keynote of the near future.

Time Is a Priceless Element

Yet here, as in the case of expanded field operational tests, it is advisable to strike a balance between today's and tomorrow's requirements, to take some steps now toward tomorrow's requirements to pick up some of that priceless element under emergency requirements—time. There must be frequent and definite points in the development of rocket engines at which production of useful articles could be started if necessary. This country must never be caught in an emergency with nothing but partially completed projects on its hands. A project that is years away from production is, from a military utility standpoint, a partially completed project. Today's gross requirements for any rocket engine are in tens; requirements in thousands may exist on short notice.

There can be no general argument with the principle that production is

both a science and an art. The science portion consists of the tools, the materials, the facilities, the methods. Given all of these, however, production is not achieved until the art of using them is developed in a team of men—and the only way that an art is learned is by practice. Furthermore, from the practice of production come the new ideas for tools and methods.

It would be uneconomic to achieve the practice of production by turning out hundreds of rocket engines for which there are no current requirements. The same would be true for complete simulation of high production practice to a point just short of actual fabrication of quantity units; i.e., complete design, operation planning, tool design, and tool procurement—for the obsolescence factor under the current development rate is high. The balance point between production for production's sake and no production practice, which is a dangerous position from a security standpoint, should approximate the following:

1 More emphasis on design from the standpoint of producibility in the course of development of those rocket-engine models for which service tests are planned. The large majority of aircraft and engines used in the last war were well along in development and initial production before we actually entered the war. Even then it took years to convert our peacetime industry to wartime production. It is generally agreed that the next time will not give us the same opportunity to prepare for combat. It follows that it is mandatory that we be ready at least to increase sharply rates of production on short notice. Every design of a rocket engine intended for field tests should be suitable for primary mass production instead of a dead-end alley on the road to high production.

2 The requirement that the components of a rocket engine which are most critical from a high-production standpoint be the subject of a detailed development for high production. This development should include design, tool design and construction, actual fabrication, and tests of some parts using these tools. Through this means the rocket-engine industry would be able to supply the design for tools and the operation plans with which qualified manufacturers outside the industry could most rapidly get rocket engines into production. It would be unusual to find even a planning concept of such tools in the industry today.

It would not be absolutely necessary to undertake such complete development for the critical component assemblies of every engine model. Once such a development program were carried out for a component assembly, future production developments on that type of component as used on other engines might well be restricted to design and tool design provided, of course, that the basic design of the component were not essentially changed. This portion of the program should be on a parallel basis with primary production of service test units to avoid undue delay in delivery of such units.

3 The occasional placement of orders for a hundred or more of a given type of engine, even though the number may exceed current requirements, so that actual full practice of production at higher rates may be achieved.

Subcontractor Group Needed

In addition to developing a production team, such a program would make possible the preliminary development of a subcontractor group, which will be of vital necessity to the rocket-engine industry when high output is re-

quired. The basic design of a rocket engine today lends itself wonderfully to the full utilization of subcontract manufacture. Apart from the amount of precision welding required, there are few fabrication techniques involved that cannot be performed by any high-grade machining and assembly shop, with a little actual experience for familiarization with the characteristics of the materials and processes used in rocket engines.

The precision welding is a different picture. At present there are only a few companies capable of handling such work to the present standards, and the services of these are most likely to be in great demand by a number of industries in an emergency. Such skills must be developed and maintained in dependable sources of supply for the rocket-engine industry, or automatic welding tools must be developed, if this processing is not to become the bottleneck of rocket-engine production. Through the tool development and sizeable production-order phases I have mentioned, this probability can be minimized.

However, if this whole program is to be effective, it must be on a continuing basis. The industry cannot expect to attract or retain personnel to build a team as a basis for rapid expansion if production or simulated production is going to drop to mere jobbing levels.

This production program is fundamentally a problem involving time and money. I believe that the responsibility for the direction of this emphasis should rest with the Armed Services. The time element is entirely under their control; the money element must be made available to them.

While it should be a fundamental objective of the rocket-engine industry to develop articles of the highest possible quality for the Armed Services, it should also be the objective of the industry to insure that developed articles have producibility. In the pressure for high performance and low weight that characterizes the rocket-engine-development field today, two fundamental factors affecting producibility are being somewhat overlooked. The first of these is the availability of critical materials in event of a national emergency. There is little value in having "available" equipment which has been developed and designed to use excessive quantities of such critical materials as nickel and chrome and the alloys of these materials. Specific contract provisions must provide in development and design phases for the reduction of such material requirements through substitution of less critical materials, if not during the course of original development, at least in a subsequent development.

The second factor affecting producibility that is in the background might be called "design beyond requirements." This affects not only material requirements, but also the most effective utilization of industrial facilities and the labor force. It is my opinion that a relatively sharp line of demarcation should be drawn between designs for rocket engines to be installed in piloted aircraft and designs for rocket engines for one-shot missiles. Obviously a rocket engine to be installed in a piloted aircraft must be in accord

with the best of aviation-design standards and incorporate the best that money can buy in safety and reliability provisions. But are the same standards necessary for the one-shot missile engine? Their use for one-shot missile rocket engines materially reduces the producibility of those engines and increases their costs.

Lack of One-Shot Liquid-Rocket Engine

Because of the development stage of missiles and their accessory equipment, the one-shot liquid-rocket engine does not exist today. In the process of making more certain that all elements of the missile will function properly in flight, the rocket engine is operated several times before free flight. It is customary to hot-test the engine before it leaves the manufacturer. At least one more static firing with the missile held captive normally takes place before actual flight. Thus, minimum anticipated operation of an engine without any margins is three times the basic minimum requirement. With normal margins, total built-in life is even higher.

While it is necessary from an economy standpoint that development rocket engines and experimental rocket engines for development missiles continue to have such high gross margins of operating life, the continuation of such margins into primary production is an expensive proceeding whose value is questionable. It is not difficult to visualize the increases in producibility and the reductions of unit costs of missile rocket engines where, for example:

1 Cold-rolled steel or aluminum is substituted for nickel and stainless steel.
2 Stampings and rolled sections are used in place of machined tubing and parts.
3 Brazed assemblies are used in place of welded, screw, and bolted joints.
4 Overhaul provisions are eliminated from most of the basic engine, which eliminates some of the interchangeability requirements.
5 Propellant valves are replaced by frangible disks.
6 Solid chambers or lined chambers are substituted for the present regeneratively cooled thrust chamber.
7 The practice of hot-testing every engine is eliminated.

This last item, the elimination of hot tests on each engine, involves the substitution of simulated operation, or "cold tests" on each engine, with occasional spot hot tests to insure that the production and simulation processes are within the predetermined control limits. Simulated operation is usable for liquid-rocket engines because operation of the engine is primarily a function of pressures and flow rates whose characteristics and limits are determinable during the development of the engine. Mechanical, i.e., moving, parts and electrical parts of the engine are small in number, and acceptability of most of these parts has long been on the basis of simulated operation. It should be remembered that the solid-propellant rocket, which is installed in piloted aircraft, exemplifies the practicality of per-

formance control through process control and spot operational checks. The technique of simulated engine operation, while not an accomplished fact for all engines being fabricated by my company, Reaction Motors, Inc., has already been used to a practical degree to minimize operational troubles occasionally encountered in hot tests of experimental production engines. Reduction of hot tests not only reduces costs in a production program but also offers some relief from the shortage of hot-test facilities.

When all of the factors of rocket engines are taken into consideration, the design standards and design specifications for rocket engines must be established to effect combinations of: (1) The status of the unit, i.e., experimental or production; and (2) the application of the unit, i.e., one-shot operation or repeated operation. Under such standards and specifications, producibility in the full sense of the word can be achieved for any rocket engine.

The problems of economics that I bring to your consideration might well be called the problems of economies, since more of them concern the use of what we have than what we do not have.

Government Is Only Customer

The development of rocket engines requires facilities, particularly those necessary for development and evaluation testing of full-scale components and full-scale engines, and for the research work in rocket-engine characteristics, fuels, and materials. Most of these facilities are specialized in that they will have little application for research and development in other fields. Furthermore, the commercial application of the rocket engine is so far in the future that to all practical purposes the Government is the only customer for rocket engines. Finally, the earnings from rocket-engine work do not, in general, support any appreciable investment in such facilities.

It is logical under these conditions that the Government bear the full burden of financing these special high-risk facilities, which it has been doing to the degree possible under present regulations and appropriations. The restrictions in existence with regard to utilization of defense funds for new facilities were made, I believe, to restrict expenditures for factories, office equipment, and general-purpose machine tools, all of which the Government owned in quantity from the last war. I do not believe that these restrictions were directed toward the special requirements of a new industry which is of great importance to the defense effort.

To make existing facilities and appropriations go as far as possible, there is a tendency to centralize test facilities for rocket engines under Government direction and administration, and to allow contractors to utilize those test facilities on a scheduled basis. For full-scale complete engine tests of the larger rocket engines, this appears to be a proper trend, for the cost of rocket-engine test stands increases terrifically with increases in engine thrust. Large engine test stands are very expensive.

For full-scale component development and basic research work, however, operations on such a basis would be most uneconomical. Apparent savings in facilities costs will be more than offset by increases in development costs resulting from conducting such fundamental operations away from the contractor's base plant.

If rocket-engine development is to proceed on a sound basis, appropriations and regulations must be established by the Government to assist rocket-engine manufacturers in obtaining the needed test facilities at their plants.

Military-Industry Team-Play

Team-play by industry and the military is the best way to secure the achievement of military articles in being. The co-ordination of effort that makes a team not only requires that someone call the signals, but also that the team members understand the play and their respective roles. The military is calling the play, as it must for military articles. But, I do not believe that the industry portion of the team understands sufficiently the direction in which the military is headed to make its most satisfactory contribution. Unless the desired end is known, it is impossible for the industry to make plans to assist in achieving that end with the greatest economy. Joint military-industry meetings for planning this team-play are very much in order—both for short and long-range plans. Military security may preclude a full revelation of where or in what direction we are expected to go. A partial revelation, however, is better than none.

Throughout this discussion, money has been mentioned several times, usually in connection with the scarcity of that element. More money is not always the only answer to an apparent scarcity—changes in the pattern of expenditure frequently afford the relief required.

For the funds available for rocket-engine research and development, it appears there may be too many activities, both military and industrial, engaged in that field for the economic good of the Armed Services. Spreading present funds over too wide a base can have the effect of curtailing the development progress in a field vital to the country's security. A reduction in the number of activities engaged in development of rocket engines might well result in a greater output of better engines, through reduction of multiplicity of effort and more intensive concentration of funds on development and producibility. Is it not worth while to take the effort to ascertain whether or not rocket-engine research and development are spread too thin?

The determination of the economic number of activities should revolve around the maintenance of competition, and in particular design competition. To any given development problem, there are usually only a few basic methods of solution. To secure more answers to the problem than the number of basic solutions does not increase design competition. It may

(*continued on page 125*)

A LOW-COST, 16-MM CAMERA FOR ROCKET PHOTOGRAPHY

By Roland E. Mueser[1] and Thomas F. Irvine, Jr.[2]

Surplus military GSAP (Gun Sight Aiming Point) cameras available to the public for as little as $20 may be modified for semihigh-speed photographic analysis. Alteration of governor and shutter makes possible film-transport speeds of over 100 frames per sec and exposure times of less than one thousandth of a sec. For a particular driving voltage, the speed of the cameras is constant within 1 per cent under controlled conditions; nevertheless a simple modification permitting a superimposed time trace is considered highly desirable. The resulting instrument is an inexpensive, durable, highly flexible semiprecision unit well suited for many types of photographic analysis.

LABORATORY and field experiments requiring medium or high-speed photographic analysis are often prohibitively expensive. Most high-speed, 16-mm cameras cost more than $1200 and are prodigious consumers of film. An inherent characteristic of such equipment is a rotating prism which makes necessary long focal lengths, hence narrow fields, and also results in slightly diffused picture images. Only a few of the standard amateur 16-mm cameras are capable of speeds of 64 frames per sec, and they are difficult to modify and cannot be readily electrically controlled. Researchers on limited budgets or those requiring large banks of medium high-speed cameras for simultaneous recording have, therefore, a difficult instrumentation problem on their hands.

The military forces in World War II utilized a specially designed 16-mm camera for aerial combat confirmation. These GSAP (Gun Sight Aiming Point) cameras are now available through war surplus dealers at prices as low as $19.50 per unit. Some retailers have even specialized in modifications to make such GSAP cameras suitable for amateur photography. It was found that relatively minor changes to GSAP cameras allow their application as rugged, flexible scientific instruments eminently suited for taking rapid picture data at low cost.

Camera Modification

The unmodified GSAP camera as received from surplus appears as in Fig. 1. To lower the current drain and make room for a timing lamp,

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internal components can be removed. These include the heating circuits which keep the camera warm at high altitudes and a relay which serves the double purpose of marking the film at the beginning of a run and limiting the number of runs. The cameras were manufactured by different concerns and differ markedly in internal construction. The authors have found the most suitable models to be those manufactured by Bell and Howell Company, and the discussion is confined to this type.

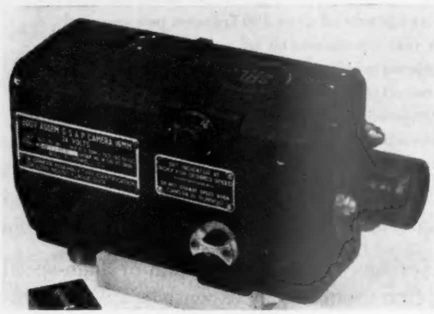

FIG. 1 UNMODIFIED GSAP CAMERA

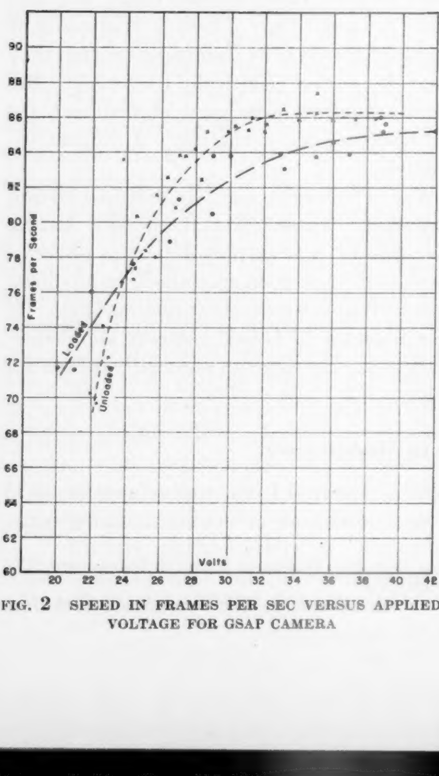

FIG. 2 SPEED IN FRAMES PER SEC VERSUS APPLIED VOLTAGE FOR GSAP CAMERA

The rotating shutter on these cameras has a 135-deg opening. Thus:

$$X = \frac{S}{360\,F}$$

where

X = exposure time for a single frame
S = shutter opening in degrees = 135 deg for cameras as purchased
F = film transport speed (frames per sec)

The standard GSAP camera as purchased has a maximum speed of 64 frames per sec and an exposure time of about $^1/_{170}$ sec. Since the shutter is a flat disk riveted in place, it may be removed and replaced with a shutter having a smaller opening. So that exposure times could be varied, new shutters were cut from 0.005-in. phosphor bronze with angles from 135 to 30 deg. They were attached by 0–80 machine screws so as to be readily interchangeable. A distortion effect due to exposing only a portion of the frame

at a time with a sweeping shutter is emphasized when using smaller angle shutters.

The cameras are powered by a small electric motor controlled by a centrifugal governor which limits operating current. Thus, the speed is adjustable and may be increased as much as 70 per cent. Fig. 2 and 3 are plots of camera speed versus applied voltage. When operating GSAP cameras at higher than rated speeds, overvoltage insures positive starting, fast speedup, and allows operation on the flat portion of the curve where motor speed is not so dependent on voltage. The cameras get quite hot when run at the highest speeds for more than 5–10 seconds at a time, although uninterrupted operation for the full length of a 50-ft magazine has been found not to be harmful. Cameras are noisy but have stood up well.

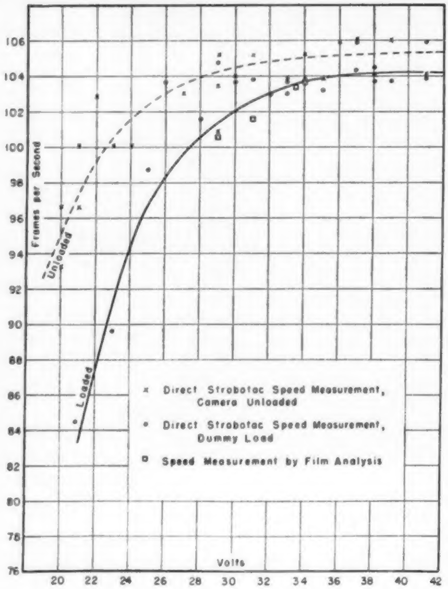

FIG. 3 SPEED IN FRAMES PER SEC VERSUS APPLIED VOLTAGE FOR GSAP CAMERA

Fig. 4 shows test films taken of high-speed calibrated clocks rotating at

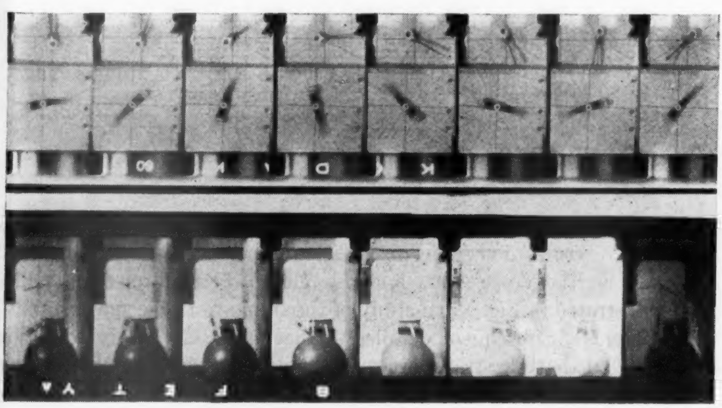

FIG. 4 GSAP CAMERA FILM-SPEED TEST STRIPS INCLUDING FLASH BULB IGNITION

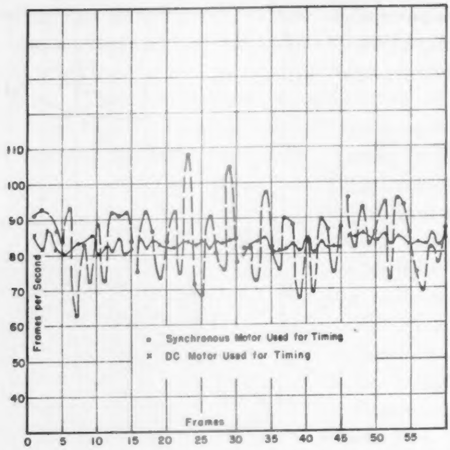

FIG. 5 APPARENT FRAME-BY-FRAME VARIATION IN CAMERA SPEED

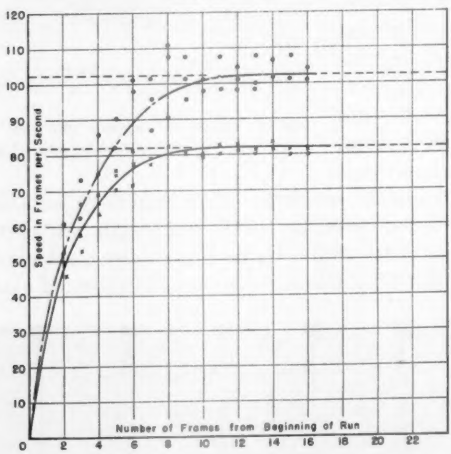

FIG. 6 AVERAGE SPEEDUP CHARACTERISTICS ON TWO GSAP CAMERAS

450 and 3000 rpm. The cameras were calibrated from photographs as well as by direct stroboscopic observation with and without film load. That a low inertia synchronous motor should not be used for time calibrations is illustrated in Fig. 5 in a study of the frame-to-frame speed stability of the cameras. Subsequent detailed studies under laboratory conditions have indicated that film-transport speed fluctuation gives a frame interval error of 1.2 per cent.[3]

[3] Probable error equivalent to 0.675 standard deviation.

The rapid speedup characteristic of the cameras is one of their chief advantages since it allows many short sequences to be made on one 50-ft magazine. Fig. 6 shows accelerations on two GSAP cameras measured in five tests.

Although limited laboratory tests with controlled voltages have achieved good speed stability, this is not a dependable characteristic. For high accuracy and field use it is mandatory that a time signal be superimposed. In the space originally occupied by the marking relay, it is possible to mount a "grain of wheat" neon bulb embedded in silvered lucite which directs a timing signal against the film edge as it appears in the feed slot of the magazine. Fig. 7 shows this arrangement in place in a camera having a 30-deg shutter.

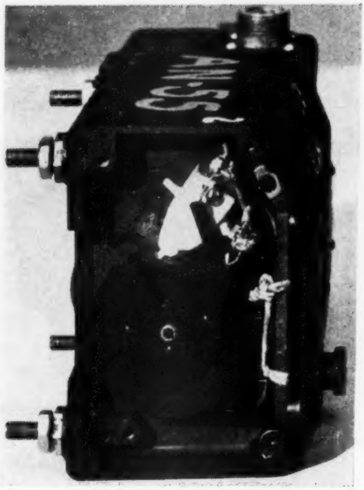

FIG. 7 GSAP CAMERA SHOWING FILM MARKER AND 30-DEG SHUTTER

The neon bulb may be operated directly from the exterior voltage of standard stroboscopic equipment. In the absence of such equipment, the timing voltage may be obtained from any 60-cycle line source with half-wave rectification. In this case a suitable series resistor must be included as a current limiter. Fig. 8 illustrates a film strip with superimposed time pips. If it is not desirable to install the internal timing apparatus, a time scale may be obtained by placing a flasher or clock directly in the camera field.

Retailers sell and mount optical view finders although a simple arrangement of sights, as in Fig. 9, is adequate and more convenient for rapid sight-

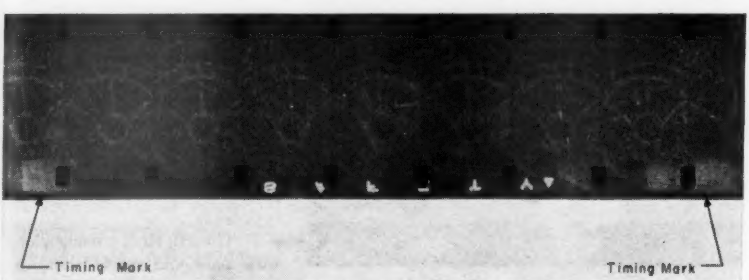

FIG. 8 FILM STRIP SHOWING TIME MARKS PLACED ON FILM BY EXTERNAL TIMER

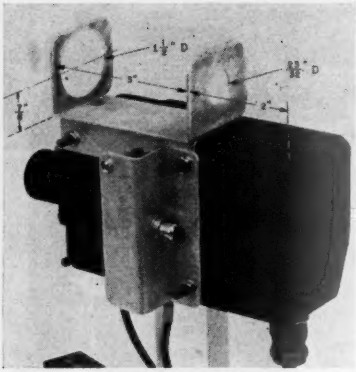

FIG. 9 GSAP CAMERA WITH VIEW FINDER FOR STANDARD 35-MM LENS

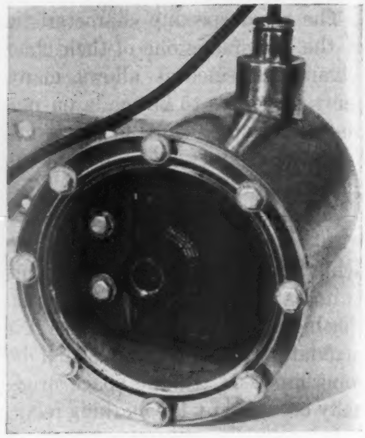

FIG. 10 GSAP CAMERA MOUNTED FOR UNDERWATER PHOTOGRAPHY

ing. The compactness of the cameras makes them ideal for underwater operation, and a model in waterproof case is illustrated in Fig. 10.

Application

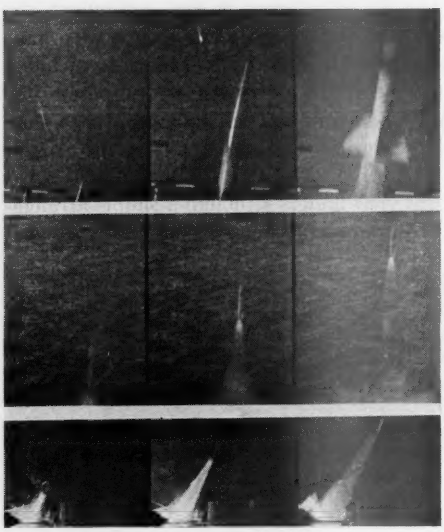

FIG. 11 PHOTOGRAPHS OF HIGH-SPEED ROCKETS MADE WITH GSAP CAMERAS

These cameras are uniquely suitable for many scientific applications. Some suggested uses are:

1 Velocity and acceleration analysis of moving objects
2 Trajectory studies of medium speed missiles
3 Remote monitoring where personal observation is impossible
4 Simultaneous recording from many angles
5 Recording from within equipment where space or weight is limited
6 Applications in situations where destruction of apparatus is possible
7 Growth studies using a long time base and single-frame exposure
8 Time and motion studies

Some samples of field photographs taken with modified GSAP cameras are shown in Fig. 11.

Characteristics

1 Speed range: 16, 32, 64 frames per sec, top speed may be increased above 100 frames per sec.

2 Practical range of shutter openings: 180 to 20 deg or less.

3 Exposure time: max at 16 fps of about $^1/_{30}$ sec; max at 64 fps of about $^1/_{130}$ sec; min at 100 fps of about $^1/_{1000}$–$^1/_{2000}$ sec.

4 Lens: standard units equipped with *f* 3.5 35-mm, although 25-mm *f* 1.5 and other lenses are available.

5 Operating conditions: 24–40 volts, 1.2–3.5 amp.

6 Film: 50-ft film magazines with Super XX film are available for $3.50. Used magazines may also be loaded by hand from bulk film.

Acknowledgment

The authors wish to thank P. G. Couperus and Hayden Schofield for their aid and co-operation in carrying out experiments with these cameras.

Problems Facing the Rocket Industry—Horne, Jr.

(*continued from page 118*)

be an aid in evaluating the methods of solution. But how much is that worth?

After the determination of the economic number of activities should come the more delicate problem of determining what specific activities, again both military and industrial, shall be among the chosen. Performance: past, present, and potential, should be the basis for this evaluation. And the evaluated performance should be in the field of rocket engines. I think it has already been ascertained that excellence of performance in one field does not necessarily mean excellence of performance in all fields.

At the beginning of this discussion it was pointed out that the rocket-engine industry is very young and very small. The problems facing this industry are not basically new problems, although they do have some conditions about them peculiar to the rocket-engine industry.

If the industry is important in our defense activities—and I think it is—the proper atmosphere for its healthy growth must exist. Mere size is not important at this time. The industry is still in its formative period. Its direction of growth is dependent in a large part on the actions taken on the problems presented here.

Action on these problems is required—and soon. But before action should come some serious thought—for the actions taken may well determine the course for the industry in its defense role for some time to come.

THE TURBOROCKET-PROPELLANT FEED SYSTEM

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Introduction

THE acceptance of the liquid-rocket engine as a practical method of flight propulsion has taken place only during the past decade. Concentrated effort in this country during the past five years has produced several engines which are currently being used effectively in supersonic flight and high-altitude research programs. The rapid development in this field has already obsoleted some engine models, and made higher altitudes and longer flight ranges possible in future applications. The liquid-propellant feed system, commonly known as the turbopump, has played an important part in these developments by reducing the total installation weight of the liquid-fueled rocket type of power plant.

Although turbopump design is based on well-founded theories, there is much to be done by the rocket-development engineer toward adapting these theories and practices to this unique application. This paper will attempt to explain the importance of exploiting conventional hydrodynamic theories and pump and turbine practice in order to achieve the desired characteristics of a turborocket engine, namely, extremely high power at a minimum dry weight.

The first part of this paper will present general turbopump systems and flow diagrams. The second part will deal with factors involved in turbopump design, followed by a third section discussing operational problems.

Purpose of the Turbopump

For those who are uninitiated in the field of the turborocket engine, it might be well to explain that the purpose of the turbopump is to supply the thrust cylinder or rocket motor with the desired amount of propellants at injection pressures required for the various thrust ratings. Pumping units are not used in all liquid-rocket applications but only in those of higher thrust and duration, where the weight of the additional machinery and fluid consumption is less than the weight of the alternative high-pressure gas-feed system. The rocket turbopump is comparable to the turbine, air compressor, and engine-driven fuel pump of an aircraft turbojet, the major difference being that the rocket turbopump absorbs only about 2 per cent of the available gas energy, while the turbojet's pumping power is usually over 50 per cent of the available energy. This is due to the fact that about 98 per cent of the turbojet's working fluid is air, requiring extremely large

volume flows and hence high hp requirements and compressor weight. Thus, the advantage of pumping only liquids to high pressures is an important factor in achieving the extremely low specific weights possible with the turborocket engine.

Although the turbopump unit is generally considered to be an accessory to the thrust cylinder, in some recent designs the turbopump assembly assumes greater complexity and weight than the thrust cylinder. This characteristic has resulted from the desire for more efficient rocket engines, necessitating higher combustion pressures, with a corresponding increase in turbopump output and size, while thrust cylinder dimensions actually decrease. Hence, the importance of further development of turbine-driven pumps to achieve high-speed low-weight designs consistent with the high-pressure thrust cylinder is one of the biggest problems confronting the turborocket designer today.

Possible Turbopump Cycles

The turborocket engine can comprise many possible arrangements of turbines, pumps, and thrust cylinders. The German engines and some of the early American designs have employed the same basic engine cycle. In this cycle the turbine working fluid is supplied by a separate energy source such as the decomposition of hydrogen peroxide. A variation of this cycle is the use of the reaction products of the engine fuel and oxidizer to drive the turbine. These gases are generated in a separate combustion pot supplied by fuel and oxidizer from the discharge side of the pumps. These two engine cycles are diagrammatically illustrated in *A* and *B* of Fig. 1.

Use of a third propellant, such as hydrogen peroxide or hydrazine, has the advantage of simplicity of controls and of reliability, since the maximum temperature is limited by the nature of the chemical reaction. However, the logistics and handling problems of such monofuels usually outweigh the system advantages where any large-scale operation of this type of engine is anticipated. For experimental upper-atmosphere sounding rockets or supersonic research aircraft, the hydrogen-peroxide turbine has proved to be quite satisfactory.

Also under this same category are those systems which depend on other separate energy sources, such as ram air or gas-turbine-compressor bleed air.

Tactical aircraft applications place more emphasis on propellant problems and ease of handling; hence, they usually employ the combustion-pot cycle regenerative turbine using the thrust-cylinder propellants. The advantages of the above two types of turbopump cycles (*A* and *B* of Fig. 1) using a separate combustion or decomposition chamber for turbine drive can be itemized as follows:

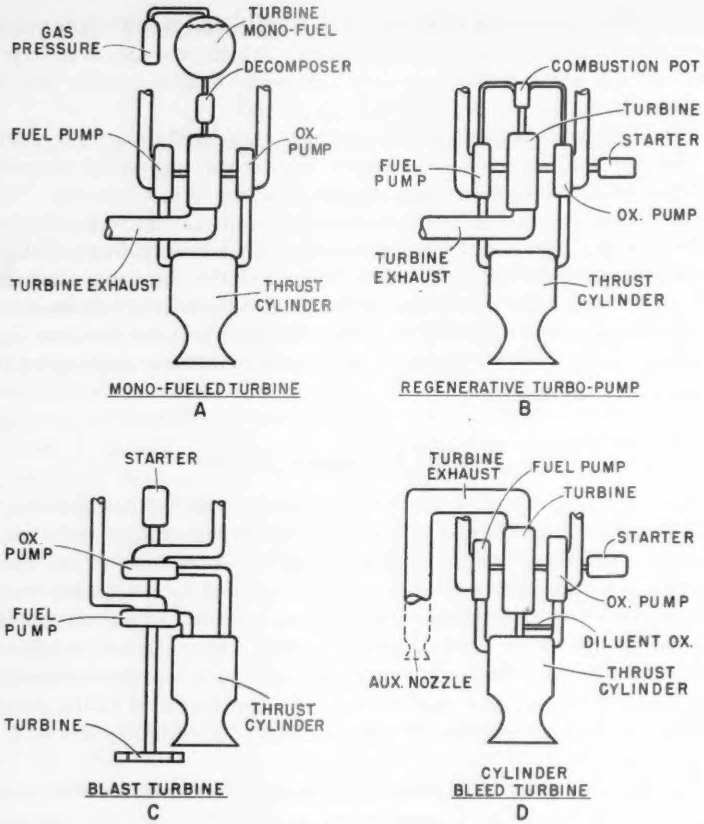

FIG. 1 TURBOROCKET CYCLES

1 The location of the turbopump unit is independent of the thrust-cylinder assembly, thus permitting considerable flexibility in airframe installations.

2 The weight of propellant tankage and gas-supply systems can be minimized by locating pumps at or near the tank outlets, thereby reducing the pressure drop in the pump suction lines.

3 The turbopump can be considered to be an independent unit capable of feeding a range of thrust cylinders operating on a given propellant combination with only minor modifications.

In contrast to these separate turbopump arrangements, there are numerous other types of more integrated turborocket cycles, which permit a considerable degree of simplification and, in some cases, an efficiency improvement. The most common of these is known as the blast-turbine cycle

illustrated in C of Fig. 1. An airfoil type of turbine is immersed in the rocket jet absorbing sufficient power to drive the propellant pumps. Control can be established by varying the depth of wheel immersion. The drag on the turbine wheel imposes a loss in engine specific impulse comparable to the gas-generator turbines. However, blast-turbine weight is usually somewhat less due to the high kinetic energy of the thrust-cylinder jet. The necessity for turbine wheel cooling either by atmospheric air or propellant vapor may result in a practical cycle efficiency slightly inferior to other arrangements.

D of Fig. 1 illustrates a second integrated cycle which is an evolution of the cycle in B, brought about by the desire for simplification. This cycle bleeds off a small portion of the thrust-cylinder combustion gases, diluting them with one propellant to achieve turbine operating gas temperatures. The advantages to be gained in arrangements C and D are:

1 Elimination of a separate turbine ignition and combustion system.

2 Integration of the turbopump and thrust-cylinder assembly into a compact propulsion unit.

A variation of the bleed cycle is also illustrated in D; it consists of recovering some of the energy normally lost due to turbine inefficiencies by means of an auxiliary thrust nozzle or nozzles. This auxiliary thrust can be conveniently used for flight-controlling purposes in some rocket installations. This cycle is particularly applicable to high chamber-pressure engines where the energy per lb of gas available to the turbine is very high. Since a lightweight turbine design can only absorb a small portion of this energy, the auxiliary nozzle cycle usually results in an improved over-all specific impulse by recovering a portion of the abnormally large turbine losses. This cycle also simplifies the control problem as the turbine becomes insensitive to back pressure variations with changing altitudes. A portion of the turbine-exhaust gases are also at sufficient pressure to be piped to the propellant-supply tanks to maintain the desired pump-inlet pressure in lieu of a separate gas-supply system.

No one of the above engine cycles is an optimum for all flight applications. Each has its advantages and disadvantages. For example, the assist take-off engine and the high-altitude sounding missile each have different requisites regarding weight, efficiency, and flexibility of component installation.

In concluding this discussion of possible turborocket cycles, let us analyze the energy balance of a typical two-propellant bleed-cycle system. Fig. 2 illustrates this energy distribution in a turborocket engine operating at 20- and 60-atm chamber pressure. A considerable increase in the available jet energy could be obtained by operating at 60 atm as illustrated by the dotted lines showing the approximate effect of the increased combustion pressure on the energy balance. The pump horsepower requirements are

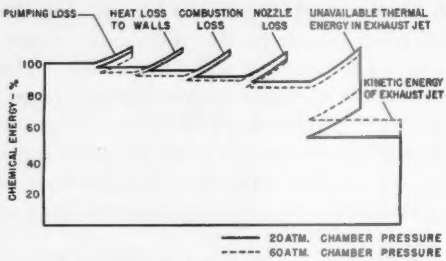

FIG. 2 TURBOROCKET ENERGY BALANCE

more than tripled by this change, but the turbine available energy is also increased. The net result is an approximate doubling of the engine pumping losses. Thrust-cylinder nozzle losses are also influenced somewhat by the higher jet velocity. However, the increase in exhaust-jet energy more than offsets these losses, resulting in an appreciable net gain. This efficiency gain, however, must be modified by the additional weight of the propellant-feed system to create this high combustion pressure.

To reduce the pumping-system weight penalties of a high-pressure turborocket engine, higher rotational and peripheral speeds are necessary. This is in lieu of resorting to multistaged pumps and turbines which can result in unwieldy and complex machinery. The short continuous duty cycle and service-life requirements are conducive to highly stressed rotating elements. Unfortunately, there are other limitations to high speed as will be discussed later.

Factors Affecting Turbopump Design

Now that we have covered the generalities of turbopump cycles and configurations, let us turn to some of the factors which influence the design of the turbopump.

The most important design parameter is the rotational speed of the turbine and pump shaft. The rpm selected determines to a large extent the weight and space envelope of the unit for a predetermined turbine consumption and pump head. The maximum turbine rpm for a fixed blade speed, and hence the minimum diameter, is limited only by its ability to handle the required gas flow. However, since the gas energy is high and the flow requirements small, it is theoretically possible to produce the required pump power by a very small high-speed turbine in the lower thrust range. The pumps, on the other hand, are restricted in speed by cavitation characteristics, necessitating a choice to be made between either the insertion of reduction gears between the turbine and pumps or the design of a slower than optimum-speed turbine on a common shaft with the pumps. The latter method results in a lighter but usually less efficient pumping plant. Here again, the flight application establishes the relative importance of the parameters of weight and efficiency influencing the design appreciably.

Pump-Design Considerations

First let us consider the effect of pump-design considerations on the

selection of shaft speed. Having determined the largest pump-volume flow, it can then be established by pump-cavitation theory that the allowable rpm varies as the three-quarters power of the available net positive suction head (*NPSH*) to the pumps. Hence, the pump-inlet pressures play an important part in establishing shaft rpm.

The establishment of this optimum pump-inlet pressure requires careful consideration by both engine and airframe manufacturer. Propellant-tankage design, propellant vapor pressure, and flight-acceleration forces are important factors in determining the available suction head. After establishing this value, the rpm can then be determined by the suction specific speed, S, formula based on pump affinity laws at conditions approaching cavitation.

$$S = \frac{rpm\sqrt{gpm}}{H^{3/4}}$$

Values of S considered feasible in conventional pump practice are in the range of 5000 to 10,000. However, the special application of centrifugal pumps to rocket engines permits a considerable increase over these values. This is due to the less critical effects of impeller pitting, noise, and efficiency loss when operating in the local cavitation region. Values of S at the limiting cavitation condition can be established by test for each propellant over a range of specific speeds. The use of a double-suction impeller (two impellers back to back) increases the allowable suction specific speed and hence the rpm by a factor of 20. The weight-saving and turbine-efficiency improvements brought about by higher allowable speeds often warrant a double-suction design in large-thrust rockets. The high vapor-pressure propellants, such as hydrogen, oxygen, and ammonia, tend to exhibit lower values of S for a geometrically similar pump than do normal liquids. This is particularly true when operating at low specific speeds where internal friction and recirculatory losses are large. This variation in suction specific-speed values with different propellants can be decreased by careful control of internal pump leakage.

Rocket-propellant pumps to feed-thrust chambers ranging from 500 to 50,000 lb of thrust cover the entire spectrum of centrifugal impeller types. Positive displacement pumps are often considered for the low-thrust applications due to their higher efficiency in this region. However, the centrifugal type has been favored in the majority of designs, because of the low viscosity and poor lubrication characteristics of most propellants.

Having determined pump-inlet pressure, outlet pressure, capacity, and maximum feasible operating rpm, a pump specific speed

$$N_s = \frac{rpm\sqrt{gpm}}{H^{3/4}}$$

can be established at its rated condition. This value is a type characteristic

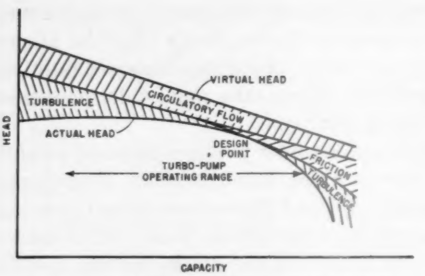

FIG. 3 PUMP HEAD CAPACITY DEVELOPMENT

of an impeller, and refers to the speed at which 1 ft of head is produced at 1 gpm. It is a measure of the general impeller proportions and characteristics, and is one criterion for predicting pump efficiency.

For variable-thrust rocket engines, it is extremely important to match the head-capacity curves for the oxidizer and fuel pumps in order to facilitate the mixture-control problem. Flat-head capacity curves are also desirable in order to minimize shaft rpm changes at various thrust levels. This affords a more rapid control-system response in addition to maintaining turbine operation close to its design point. Control of the head-capacity curves can be accomplished to some degree in the pump-design stage, by vane shape, selection of the shockless entry point of impeller and diffuser vanes, and accurate estimation of friction and recirculation losses over the flow range. Fig. 3 illustrates how the head losses are distributed over a typical propellant-pump head-capacity curve, using backward-flow vane design.

Although it is a relatively simple design problem to match the head-capacity curves of two pumps in the same specific speed range, it is rather difficult to predict the exact shape of these curves, because of the lack of knowledge concerning the nature and magnitude of the hydraulic losses. Shifting of the head-capacity curves at constant speed can be accomplished by impeller-diameter trimming or diffuser-area adjustment. The former method is usually used to correct for designing inaccuracies. Manufacturing tolerances also have an effect on the placement of the head-capacity curve. Orificing of the pump discharges is usually employed to compensate for these inaccuracies in order to achieve the desired thrust-cylinder mixture ratio. In certain cases auxiliary flow-controlling devices may be required.

Hydraulic balancing of the individual pump impeller is not so important as achieving combined balance of the oxidizer and fuel impeller when mounted in an opposed position on a common shaft. Thus, during normal operation bearing-thrust loads can be practically eliminated by a cancelling out of all hydraulic forces. It is often necessary to absorb all the turbine horsepower by only one pump for very brief periods at the point of exhaustion of one propellant. This operation can impose tremendous thrust loads on the bearing unless some balancing precautions are taken. This, however, is true only of monofueled-turbine designs where the turbine power is unaffected by pump output.

In order to achieve lighter weight and more efficient pump designs, the turborocket engineer is striving to increase the rotational speed as the required diameter of the rotating machinery bears an inverse relation to the rpm. Thrust cylinders are also demanding increased injection pressures, necessitating the development of extremely high pump heads in a single stage. Pump heads of 2500 to 4000 ft are often necessary for some of the lower-density propellants. Weight limitations often require this head to be achieved in one stage, which can be done with a reasonable efficiency only by a high rpm. A great deal of consideration has been given to the possibility of achieving controlled pumping conditions in the cavitation region to derive the weight benefits of the higher speeds. These requirements differ considerably from those used in commercial pump practice, opening up an entirely new era in the field of centrifugal pumps to achieve the extremely lightweight compact designs necessary for the turborocket engine.

Turbine-Design Considerations

Now let us turn to turbine-design considerations. The impulse turbine is nearly always selected as the prime mover in the turbopump because of its simplicity, light weight, and ability to operate at high speeds and temperatures for short periods. Like the pumps, this turbine application is unique in the fact that extremely high rates of energy transfer must take place in a minimum of stages. The high pressures feasible with liquid-feed pumps result in very high combustion-gas densities, large pressure ratios, and hence high available energies. It is conceivable to have from 250–500 Btu per lb available for conversion to shaft power by the turbine. Resulting impulse-turbine-nozzle velocities range from 3500 to 5000 fps, necessitating higher than practical blade speeds or numerous reaction stages to achieve optimum turbine efficiency. However, since the pumping energy is a very small fraction of the turborocket-cycle energy, in most applications it is not feasible to attempt to achieve high turbine efficiencies at the expense of a large weight increase. For example, the German Walter engine developed 6000 hp at 600 miles per hr with the pump-driving turbine producing only 125 hp, or about 2 per cent of the net output. In actual design practice the turbine fluid consumption and dry weight must be balanced to achieve the highest performance of the aircraft. In most rocket applications this analysis results in a turbine of one or two stages. Terry and Ljungstrom type turbines have also been considered. In general such turbines would operate in the efficiency range of 40 to 60 per cent (for atmospheric exhaust pressure) but would have an extremely low weight-horsepower ratio.

The turbine working fluid in a regenerative system usually consists of either a strong reducing or an oxidizing atmosphere, each having its disadvantages. Fig. 4 illustrates the possible operating points of the turbine

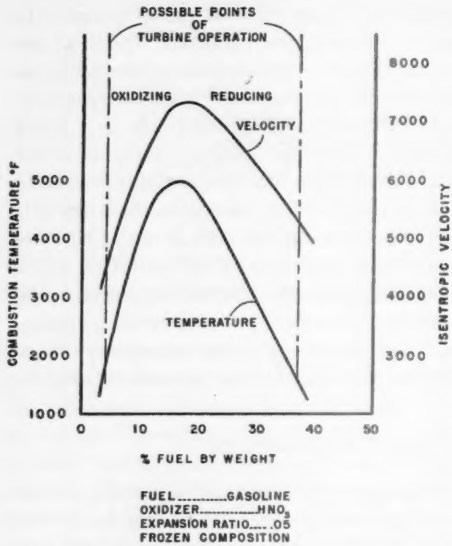

FIG. 4 TURBINE COMBUSTION POINTS

on both the temperature and gas-velocity curves, as plotted against mixture ratio. Turbine operation on the fuel rich side of the stoichiometric ratio usually yields a slightly higher isentropic velocity for the same gas temperature due to the release of free hydrogen. This results in a lower average molecular weight of the gas mixture than is produced by the oxides resulting from combustion on the oxidizing side of the curve. Turbine operation in the strong oxidizing gas is not conducive to high inlet temperatures. Nickel or cobalt-base alloys are the best oxidation-resistant materials if operation above 1300 F is anticipated. Fuel-rich operation usually produces free solid carbon which tends to clog injector holes and coat all turbine surfaces. Free hydrogen and hydrocarbons, such as methane, are also formed which can cause hazardous operation unless explosive and sealing precautions are taken.

Actual turbine-nozzle velocities are usually considerably lower than the calculated isentropic velocity because of nozzle friction losses. There is little or no published data on small high-velocity turbine-nozzle losses, since they are rarely utilized in conventional turbine practice. However, it is generally considered that nozzle efficiencies in the propellant turbine are in the range of 75 to 90 per cent. This is in contrast to the larger flow thrust-cylinder nozzle which is believed to be close to 98 per cent efficient.

The turbine design depends largely on the type of cycle selected and on the flight application. The single-impulse wheel is the lightest design but can absorb only a small portion of the available energy in atmospheric exhausted turbine cycles, because of blade speed and weight limitations. The efficiency can be improved by the addition of a re-entry chamber with only a slight weight addition, when small volume flows are involved. The Curtis stage turbine is commonly used in this application, since reasonable efficiencies can be obtained in the range of blade-to-jet-velocity ratios normally encountered.

For the longer duration flight applications where turbine consumption

is the most important consideration, methods must be developed whereby a greater portion of the available gas energy can be converted to shaft power. One such promising method appears to be in the use of a reheat cycle where the available enthalpy drop can be divided among three or four small-diameter impulse stages. Reheating of the exhaust gas from each stage can be accomplished by the injection of additional fuel in an oxidizing gas turbine, and vice versa. In this manner the available energy approaches that of stoichiometric combustion, and also more favorable blade-to-jet-velocity ratios can be achieved. With some propellant combinations reheat cycles have been investigated resulting in brake specific fluid consumptions as low as 5 lb/bhp-hr at altitude operation. This compares very favorably with stationary steam-condensing power plants which have weight-horsepower ratios many times greater than the propellant-combustion turbine. However, the majority of the turborocket applications, because of their short-flight operation, demand less complex, lighter turbines having consumptions in the range of 15–25 lb/bhp-hr. This is in contrast with the air-cycle gas turbine which utilizes in the neighborhood of 50 lb/bhp-hr of total working fluid. The much higher pressure ratios and low pumping losses in the liquid-feed turbine afford this substantial reduction in working fluid.

The application of the gas turbine to the rocket engine permits higher creep rates and, hence, higher stresses than are commonly used in turbine practice. Ten-hour intermittent operation is considered to be a maximum feasible design operating life of a turbopump unit for current applications. Since there is no warm-up time permissible in these applications, thermal shock in turbine components is a very important consideration. This is particularly true where liquid oxygen is used as the oxidizer, necessitating a cooling-down period prior to operation. It is not inconceivable that during the first 10 sec of operation turbine-rim temperatures may be 1000 F, while hub temperatures are still below zero, imposing very large thermal stresses in the turbine disk. The Germans, in V-2 turbopump design, precluded this condition by using two shafts with an insulating flexible coupling between the turbine and liquid-oxygen pump.

Current turbine designs are based on inlet temperatures in the range of 1200 to 1500 F and maximum peripheral velocities of about 1300 fps. Further work in combustion-temperature control will undoubtedly increase the inlet temperature. Blade speeds can also be raised without a weight penalty when higher rpm will permit smaller wheel diameters.

Mechanical Considerations

Some important mechanical considerations are seals, bearings, and accessory-drive gears. Sealing of high-pressure nonlubricating fluids around high-speed shafts presents one of the most difficult mechanical problems. Since the turbine case contains a hot reducing or oxidizing gas, it becomes

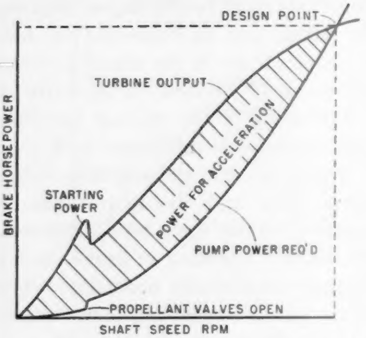

FIG. 5 TURBOPUMP ACCELERATION CHARACTERISTICS

extremely important to control both pump and turbine leakage carefully to prevent the formation of an explosive mixture within the turbopump. Sealing these fluids effectively for periods up to ten hours requires extremely precise hydraulic balancing and surface finishing. Nearly perfect hydraulic balancing can be achieved when empirical knowledge of the pressure gradient across the seal face is known, and where very accurate construction tolerances can be achieved.

Both antifriction and sleeve bearings have been used successfully in turbopumps. Both types have used the propellants as lubricants where long life is not required. Liquid-oxygen-pump bearings present the greatest lubrication problems, since all lubricants become solid before −100 F is reached. Forced lubrication systems, usually employed on high-speed shafts to remove bearing frictional heat, are not always warranted in the rocket turbopump because of the short operating periods.

Since the turbopump shaft is the only possible source for accessory power in a turborocket engine, it often becomes necessary to incorporate up to a 20-hp, lower speed, alternator and/or hydraulic-pump power take-off. This auxiliary power is used for flight-control purposes and also for telemetering data to the ground in the high-altitude sounding rocket.

Turbopump Operation

Now let us consider the operating characteristics and control problems of the turbopump unit. In operation with the constant-thrust engine, the major control problem is to adjust the turbine output to equal the pump power requirements at their design point on the rated-speed head-capacity curves. Fig. 5 illustrates typical acceleration-power characteristics of the turbine and pumps up to the point of stabilization with a regenerative type pumping system. This is actually a "boot strap" process after the initial acceleration is given to the turbine from an external starting source such as an electric motor, pressurized starting propellants, powder charge, or air blast. The shaded area in Fig. 5 represents the power available for acceleration which can be integrated into an rpm-versus-time curve. The accelerating time can be controlled by varying the starting horsepower or point of opening the propellant valves, both of which affect the shaded area. The final adjustment of the stabilization point is usually made by

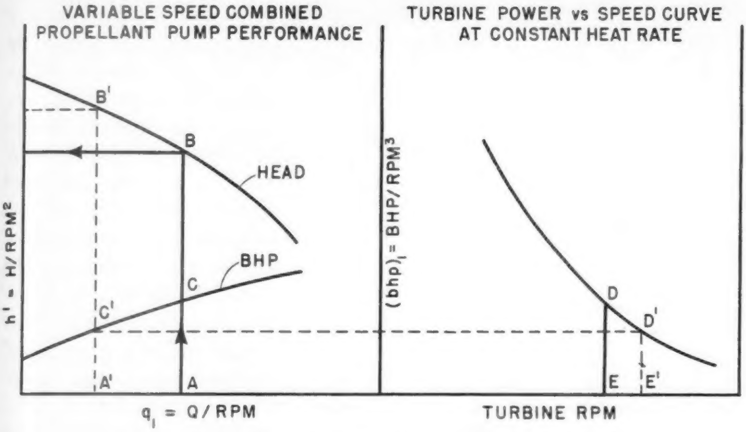

FIG. 6 DETERMINATION OF TURBOPUMP OPERATING POINTS

orificing the turbine propellants, or by increasing the turbine-nozzle-throat area tending to lower or raise the stabilized point.

Determination of the operating points of a turbopump feeding a variable-thrust rocket is considerably more complex, since every change in thrust or flow demands results in an rpm shift to the point where the turbine and pump brake horsepowers meet. Determination of the operating point for any flow condition requires knowledge of the variable-speed head capacity curves of the propellant pumps operating in parallel, and also the horsepower-versus-speed characteristics of the turbine driven by a constant gas-energy source. This method makes use of the unit capacity, $q_1 = Q/\text{rpm}$; unit head, $h_1 = H/\text{rpm}^2$, and unit brake horsepower, $\text{bhp}_1 = \text{bhp}/\text{rpm}^3$, all derived from centrifugal-pump affinity laws. Fig. 6 illustrates the combined performance of the propellant pumps plotted in terms of unit capacity, unit head, and unit brake horsepower. Also plotted on a common unit brake-horsepower scale is the turbine-power curve versus speed curve reduced to unit-speed terms by dividing power by the corresponding rpm^3. Then, for a full thrust condition corresponding to unit capacity, A, the unit head is B, pump unit brake horsepower is C, which is equal to that of the turbine D. The rated turbine speed is then at E, which is also the pump rated speed. Similar diagrams can define the actual head and capacities at any shaft speed by multiplying unit capacity by the rpm and unit head by the rpm^2. For instance, if the discharge flows were throttled to produce approximately 50 per cent of rated thrust, the operating points would be $A'B'C'D'E'$, resulting in an increase in both rpm and pump head for an ungoverned turbine driven by a separate energy source.

In most variable-thrust engines, it is necessary to regulate the turbine power to maintain constant pump discharge pressure at any flow demand.

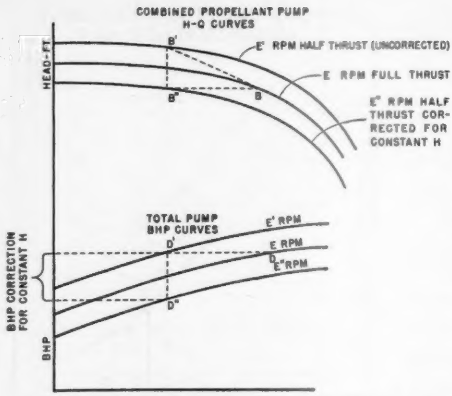

FIG. 7 CONSTANT SPEED PERFORMANCE CURVES

An analysis similar to Fig. 6 is very helpful in predicting turbine-governor requirements. For example, this ungoverned turbine results in a speed increase from E to E' at some reduced thrust. Referring to the constant speed performance curves, Fig. 7, the necessary reduction in turbine horsepower to maintain constant propellant pressures at the reduced thrust can be readily determined. This method results in a corrected speed E'' produced by a horsepower correction of D to D''. In this manner, the turbine-fuel or gas-metering system can be designed to maintain constant pump discharge pressures by proper correction of turbine power over a range of engine thrust. Regulation can be achieved by various methods. These include metering propellants, throttling inlet gases, waste gate, or back-pressure control. These governing valves are actuated by a servomechanism which is sensitive to pump-discharge pressure.

In addition to maintaining the proper fluid power to the thrust cylinder, the turbopump should be equipped to prevent overspeeding due to loss of hydraulic load, and overheating due to a shift in mixture ratio, if a bipropellant-combustion turbine is used. These two conditions are in most cases caused by pump cavitation or exhaustion of one propellant, permitting these safety controls to be overriding lockout circuits.

This paper has attempted to illustrate why the rocket engine is no longer the simplest of all engines containing no moving parts. This type of engine, in fact, employs a cycle not unlike the turbojet, but is unrivaled in horsepower per unit weight and frontal area. Future turborocket engines will utilize high combustion pressures, small high-speed turbines with over 2000 F inlet temperatures, and directly coupled pumps operating under "controlled" cavitation conditions. To achieve this type of design it is obvious that the rocket engineer cannot rely on conventional turbine and pump practice alone, but must use it as a basis to launch into a new era of much higher speed and greater energy transfer per unit weight.

Although there has been insufficient time to discuss many of the interesting details of turbopump design and development problems, it is hoped that this paper has succeeded in furthering the readers' understanding of the theory, current practice, and operational problems of the turborocket-engine propellant-feed system.

APPLICATION OF WHITE FUMING NITRIC ACID AND JET-ENGINE FUEL (AN-F-58) AS ROCKET PROPELLANTS

By M. J. Zucrow[1] and C. F. Warner[2]

Nomenclature

a = absorptivity of surface
c^* = characteristic velocity
C_T = thrust coefficient
c_p = constant pressure specific heat
c_v = constant volume specific heat
C_c = correction for effect of total pressure on CO_2 radiation
C_w = correction for effect of total pressure on H_2O radiation
D = chamber diameter
E_s' = effective surface emissivity $\approx \dfrac{E_s + 1}{2}$
E_s = surface emissivity
E_g = emissivity of gas mixture
E_{gc} = emissivity of CO_2 at gas temperature
E_{gw} = emissivity of H_2O at gas temperature
ΔE_g = correction for superimposed radiation at gas temperature
G = mass flow density
h_c = convective heat transfer coefficient
I_{sp}' = theoretical specific impulse
K = thermal conductivity
k = specific heat ratio
L^* = characteristic length of rocket motor
Pr = Prandtl number $\left(\dfrac{c_p\mu}{K}\right)$
P_c = combustion chamber pressure
q_c = convective heat-transfer rate
q_r = radiant heat-transfer rate
q_T = total heat-transfer rate
Re = Reynolds number $\left(\dfrac{VD\rho}{\mu}\right)$ or $\left(\dfrac{DG}{\mu}\right)$
T_g = gas state temperature
T_i = impressed temperature
T_s = wall temperature
T_T = gas total temperature
V = fluid velocity
μ = dynamic viscosity of fluid
ρ = density of fluid

Introduction

IN RECENT years there has been a great interest, both in the United States and in certain European countries, in the application of white fuming nitric acid (WFNA) and a hydrocarbon fuel as rocket propellants. In this country jet-engine fuel AN-F-58 has been selected as the hydrocarbon fuel. The principal reasons for the selection of these propellants are apparently related to the problems of availability and logistics. Both propellants are liquids, can be made readily available in an emergency, are relatively inexpensive, and have satisfactory freezing points. Furthermore, the performance of the WFNA-jet-engine-fuel propellants is not

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greatly different from the nitric acid-aniline or aniline-furfuryl-alcohol propellants with which a large body of experience has been accumulated. The fields of application of the WFNA-jet-engine-fuel propellants are envisioned as being those where the consumption of propellants will be large, and the operating requirements can be satisfied with propellants giving performances substantially equal to those obtainable with the nitric acid-aniline type propellants.

Calculated Performance Parameters for WFNA-Hydrocarbon Propellants

At the present time there are only limited experimental data concerning the performance of the WFNA-(AN-F-58) propellants. To obtain a background of what can be expected from these propellants, calculations have been reported by C. H. Trent and M. J. Zucrow (1, 2)[3] on the performance of WFNA and different hydrocarbon fuels. The purpose of the calculations was to determine the effects of H/C-ratio, and oxidizer/fuel (O/F) ratio upon the calculated impulse I_{sp}'. Since the combustion pressure influences the specific impulse, the effect of varying the combustion pressure from 300 to 2000 psia was also studied. The reactions of WFNA with the following hydrocarbons were studied; heptane (C_7H_{16}), octane (C_8H_{18}), decane ($C_{10}H_{22}$), and eicosane ($C_{20}H_{42}$), or a range of C/H ratios varying from 5.25 to 5.73. These hydrocarbon fuels were considered to be representative of the range of C/H ratios to be expected in the AN-F-58 fuel. The results indicated that the performance characteristics of the AN-F-58 fuel would not differ materially from those obtained by reacting octane with WFNA. Consequently, the studies presented in this paper are based on that assumption.

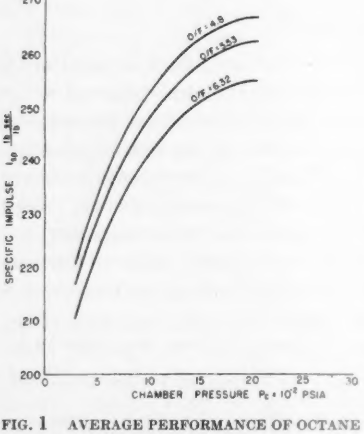

FIG. 1 AVERAGE PERFORMANCE OF OCTANE OXIDIZED BY WFNA

Effect of Combustion Chamber Pressure

Fig. 1 illustrates the effect of the combustion pressure upon the specific impulse I_{sp}' for the reaction between octane and WFNA. Three different mixture ratios are presented: O/F = 4.8, 5.53, and 6.32. As to be expected, the curves indicate that a substantial improvement in performance can be effected by

[3] Numbers in parentheses refer to Bibliography at end of paper.

raising the chamber pressure above current value of 300 psi (11). The increase in the specific impulse results primarily, however, from the larger expansion ratios available with complete expansion of the gases when higher combustion pressures are employed. In other words, the higher specific impulses correspond rather to increased values of the calculated thrust coefficient C_T' than to the slightly larger values of the characteristic velocity c^*. The latter conclusions are illustrated by Fig. 2 wherein C_T' and c^* are presented as functions of the combustion pressure.

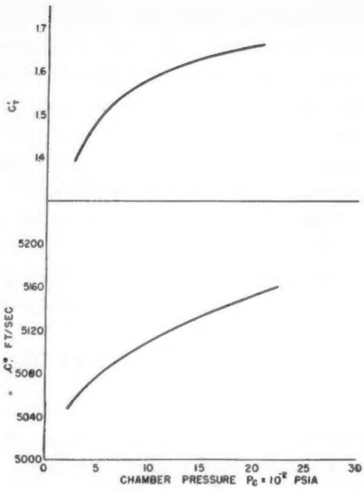

FIG. 2 VARIATION OF c^* AND C_T' WITH CHAMBER PRESSURE FOR O/F = 4.8 EMPLOYING OCTANE OXIDIZED BY WFNA

Fig. 3 illustrates the influence of the expansion ratio A_e/A_t for the nozzle upon the thrust coefficient C_T' for five different values of combustion chamber: 300, 600, 1000, 1500, and 2000 psia.

In a conventional rocket motor the nozzle has a fixed expansion ratio A_e/A_t. Consequently, as the rocket moves to different altitudes the area ratio cannot be the optimum for all altitudes so that the ultimate performance possible cannot be realized at all altitudes. The application of higher combustion pressures aggravates the expansion-ratio problem. In many applications, however, it is desirable to control the thrust developed by the rocket motor. All efforts to achieve thrust control by means of a variable-area exhaust nozzle have been unsuccessful and that type of solution does not appear promising. An alternate method for achieving thrust control is to vary the combustion pressure, but if the maximum combustion pressure is set too low, 300 psi for example, then at low thrust outputs the specific

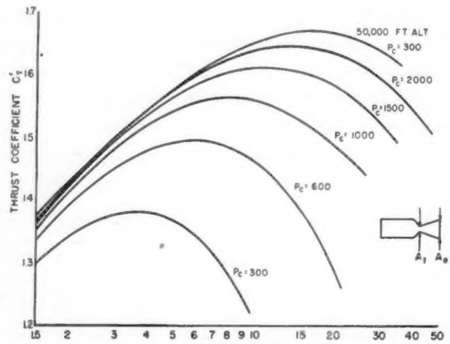

FIG. 3 VARIATION OF C WITH CHANGES IN EXPANSION RATIO FOR THE STOICHIOMETRIC MIXTURE RATIO OF OCTANE AND WFNA. ATMOSPHERIC PRESSURE ASSUMED TO BE EQUAL TO 14.7 PSIA

propellant consumption increases rapidly and the phenomenon of "chugging" is encountered. Fig. 3 illustrates that by designing the rocket motor so that the maximum thrust is secured by operating at high combustion pressure, satisfactory specific propellant consumption should be achievable over a wide range of thrust if the expansion ratio for operation at the cruising altitude is based on obtaining the cruising thrust with a combustion pressure between 250 and 300 psi.

Where the rocket engine is to be operated at a sea-level altitude, such as in jet-assisted take-off units, the selection of the most appropriate combustion pressure should be based on that which gives the lightest over-all weight, or other practical considerations.

Ignition of WFNA-Jet-Engine-Fuel Propellants

An engineering problem introduced by the utilization of the WFNA-jet-engine-fuel propellants, which are nonhypergolic, is the development of a reliable simple ignition system. While it may be satisfactory to start the rocket motor for test purposes by introducing a hypergolic propellant combination ahead of the main propellants, that method of starting the rocket motor is undesirable from several practical viewpoints.

Some experimental studies of the reactions of WFNA with nonhypergolic hydrocarbon fuels have been conducted at Purdue University, with emphasis on the reactions of WFNA with gasoline, JP-I, and AN-F-58 fuels. One set of experiments was concerned with the approximate determination of the required minimum temperature of WFNA vapor to cause the ignition of ethyl gasoline.

It was found by introducing ethyl gasoline (at room temperature) dropwise into an environment of WFNA vapor at 700 F that the combustion of the gasoline proceeded with the emission of visible light after approximately 5 sec. The ignition delay was shortened to practically instantaneous combustion by raising the WFNA-vapor temperature. It was also determined that a fine stream of gasoline is ignited as it comes in contact with an atmosphere consisting of WFNA vapors at a temperature above 700 F. It is known that nitric-acid vapor when heated to a sufficiently high temperature dissociates according to the following equations:

$$HNO_3 \rightarrow H_2O + NO_2 \qquad [1]$$

and

$$2NO_2 \rightarrow 2NO + O_2 \qquad [2]$$

Consequently, the reaction of gasoline with hot nitric-acid vapor appears to be basically a reaction between gasoline and heated oxygen at an elevated temperature. The rate of reaction is probably controlled by the rate of formation of O_2 from the decomposition of NO_2, but that is yet to be established.

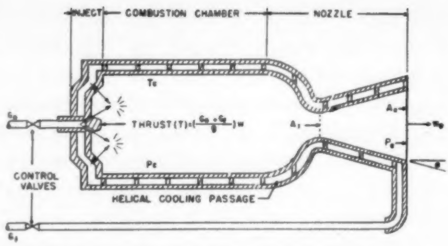

FIG. 4 PRINCIPAL ELEMENTS AND STANDARD NOTATION OF A LIQUID PROPELLANT ROCKET EMPLOYING REGENERATIVE COOLING

Experiments were also conducted wherein individual streams of AN-F-58 fuel and WFNA were caused to impinge upon a heated metal surface. It was found that a stainless steel surface at 1000 F (min) produced ignition with some ignition delay. By premixing the liquid propellants for a length equal to 1 cm and then discharging them upon a stainless-steel plate heated to 985 F, ignition occurred with no appreciable ignition delay. It was also found that by coating the heated surface with different materials the minimum surface temperature for ignition could be reduced.

These experiments indicate that it should be possible to solve the ignition problem along conventional engineering development lines.

Cooling of Rocket Motor

Fig. 4 illustrates schematically the regenerative cooling method employed in current rocket-motor designs. Considerable experience has been accumulated with that method of cooling and with rocket motors employing nitric acid as the oxidizer. Most of that experience, however, has been with combustion pressures of the order of 300 to 400 psia and with either aniline or aniline-furfuryl-alcohol mixtures as the fuel. It is of interest, therefore, to estimate the adequacy of regenerative cooling first, if the fuel is changed to jet-engine fuel (or octane), and second, if the combustion pressures at which they are utilized are increased.

Heat is transferred from the hot radiant combustion gases to the relatively cool walls of the motor by forced convection and radiation, the over-all rate of heat transfer being the sum of the contributions of each mode of heat transfer.

It is difficult to analyze the heat transfer accurately because the detailed phenomena are incompletely understood. Consequently, to estimate the heat-transfer rates it was necessary to extrapolate data based on experiments which are not representative of the conditions existing in a rocket motor. Furthermore, it should be pointed out that even if the average over-all heat-transfer values could be calculated accurately, they cannot be regarded as absolute criteria of the ability of the rocket motor to operate with "burnout." There is sufficient experimental evidence to indicate that for a given combustion-chamber configuration the design of the propellant injection system can exercise a major effect in producing local "hot spots." Because of the lack of knowledge of the exact manner in which the injection-

design and combustion-chamber configuration influences the heat transfer, a close agreement between the calculated and the experimental values of heat transfer in a specific case cannot be expected.

Despite these inadequacies of the current methods for estimating heat-transfer rates, the calculated and the experimentally determined values of the heat-flux densities for the critical sections of the motor are instructive and may be considered as preliminary qualitative information regarding the severity of the cooling problem.

Convective Heat Transfer: The conventional approach for calculating the heat transfer by convection is based on the analogy between heat transfer and momentum transfer due to Reynolds in 1874. Extensions of this analogy have been made by G. I. Taylor and Theodore von Kármán (3, 4). The resulting equations are applicable, however, to the case where the hot fluid flows in a long straight tube with fully developed turbulence and the temperature difference between the bulk of the fluid and wall is small, so that the flow is essentially isothermal; conditions which are quite different from those occurring in a rocket motor. But that approach is the only one currently available to the rocket-motor designer (5). The equation for the convective heat-transfer rate q_c can be written in a form similar to that for heat transfer by conduction. Thus

$$q_c = h_c(T_g - T_s) \qquad [3]$$

The theoretical calculation of h_c is difficult, because it is a function of the variables influencing the boundary-layer thickness, the temperature distribution within the boundary layer, and perhaps several others. Since no completely satisfactory theoretical relationship for evaluating h_c has as yet been developed, its value is determined empirically for different experimental conditions. Dimensional analysis and experimentation demonstrate that convective heat-transfer data for fluids flowing in long tubes can be correlated by expressing h_c as a function of the Reynolds and the Prandtl numbers. Such a relationship for the cooling of fluids is as follows (6):

$$h_c = 0.265\,\frac{K}{D}\,(Re)^{0.8}(Pr)^{0.3} \qquad [4]$$

Equation [4] correlates experimental data when the Reynolds number is less than 500,000 and when the temperature difference between the fluid and the wall is small. The numerical values for the physical properties of the fluid, when applying Equation [4], are evaluated at the static temperatures of the main body of fluid.

The available experimental data indicate, however, that the flow Mach number has a negligible effect upon the magnitude of the heat-transfer coefficient, h_c, if the static temperature of the gas T_g in Equation [3] is replaced by what has been named either the "impressed" or the "adiabatic wall" temperature T_t (7). The latter is defined as the temperature assumed by the wall in the presence of the hot gas flowing past it but in the absence

of heat transfer. If T_T denotes the total temperature of the flowing gas, and α the recovery factor for the gas stream, then T_i can be evaluated from the equation

$$T_i - T_g = \alpha(T_T - T_g) \quad [5]$$

Experiments show that the recovery factor α can be evaluated by Equation [7] (12)

$$\alpha = (Pr)^{0.33} \approx 0.90 \text{ (for air)} \quad [6]$$

Most of the reported investigations of heat-transfer coefficients for large fluid velocities were performed with relatively small temperature gradients between the fluid and the wall. McAdams and others have shown, however, that for air flowing at high subsonic velocities in heated metal tubes, the effective mean heat-transfer coefficient for the air is independent of the temperature difference between the bulk temperature of the fluid and the metal wall.

Humble, Lowdermilk, and Grele (8) studied heating of air flowing through a metal tube heated to a surface temperature of 2060 R. They correlated their data by means of the equation

$$\frac{hD}{K_s} = 0.23 \left(\frac{\rho V_s D_b}{\mu_s}\right)^{0.8} \left(\frac{c_{p_s}\mu_s}{K_s}\right)^{0.4} \quad [7]$$

In Equation [7] the subscript s denotes that the fluid property is to be evaluated at the temperature of the metal surface, and the subscript b that the property is to be evaluated at the bulk temperature for the fluid. The validity of correlating Equation [7] in the reverse case, a highly heated gas flowing past a cool wall, needs experimental verification.

Equations [4] and [7] are at best macroscopic approximations. Nevertheless, the authors employed them for calculating the estimated heat-transfer coefficients presented in this paper.

The evaluation of the physical properties of the combustion gases to be substituted into Equations [4] and [7] posed additional problems because of the lack of experimental data at the temperatures occurring in rocket motors. Consequently, the available data have been extrapolated.

It was assumed that the composition of the gas mixture and the molecular weights of the component gases corresponded to those obtained by the thermochemical equilibrium calculations for WFNA-octane reported in (1). The heat-transfer calculations assumed that the propellants were reacted at the stoichiometric mixture ratio.

The combustion temperatures in rocket motors are high, 5000 to 7000 F, the temperatures (for a given mixture ratio) increasing with the combustion pressures. Although the combustion temperatures are well above the critical temperatures of the individual gas components forming the combustion-gas mixture, the corresponding combustion pressures, over the range studied, are only moderate compared to the critical pressures. It

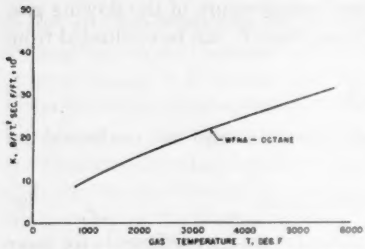

FIG. 5 THERMAL CONDUCTIVITY OF COMBUSTION-GAS MIXTURE

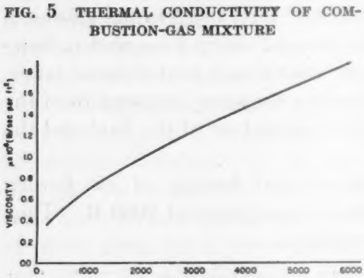

FIG. 6 VISCOSITY OF COMBUSTION-GAS MIXTURE

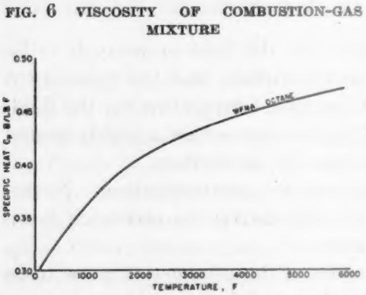

FIG. 7 CONSTANT PRESSURE SPECIFIC HEAT OF COMBUSTION-GAS MIXTURE

seemed reasonable, therefore, to assume that the pertinent physical properties for the combustion gases are, in the main, functions of temperature alone.

The values of thermal conductivity for the component gases were calculated by the kinetic theory relationship (9).

$$K = \bar{E}c_v\mu \qquad [8]$$

where $\bar{E} = {}^1/_4(9k - 5)$.

Fig. 5 presents the calculated values for the thermal conductivity of the combustion-gas mixture (based upon the weighted average of the individual values for the individual gaseous components) as a function of temperature.

No published experimental data are available regarding the viscosities of gases over the temperature range encountered in rocket motors. The Sutherland equation was employed to extrapolate the viscosities of the individual gas components—a procedure which is of course open to question. Fig. 6 presents the extrapolated values of viscosity used in the analysis. The constant pressure specific heat of the combustion gas mixture is presented in Fig. 7.

The density calculations for the gas mixture assumed that all of the gaseous components behaved as perfect gases.

In computing the convective heat-flux densities the following conditions were imposed upon the rocket motors investigated: (1) All cylindrical in shape; (2) all had a length/diameter ratio of 2; (3) all developed 500 lb thrust; (4) all had an $L^* = 100$ in.; (5) there was no change in the chemical aggregation and physical properties of the combustion gases from that for the combustion chamber; and (6) the combustion corresponded to the stoichiometric mixture ratio for WFNA octane.

Heat-flux densities were calculated for the WFNA-octane propellants for combustion pressures ranging from 300 to 2500 psi.

Radiant Heat Transfer: The contribution of radiant heat transfer was

assumed to be due entirely to the water-vapor and carbon-dioxide components of the combustion gases. The radiation from CO was neglected because of the small quantity present.

The method developed by Hottel and Egbert was considered as being the most suitable for calculating the radiant heat transfer (10). The method was developed for application to gas furnaces operating at moderate temperatures and low pressure. Because of the lack of data on the effect of pressure on radiant heat transfer, the Hottel-Egbert curves were extrapolated to the combustion pressures employed in the study. Objections can be raised to the application of that method to the problem at hand, and the authors will welcome any suggestions for estimating the radiant heat transfer in rocket motors with greater accuracy.

The general equation for the heat radiated by hot gases to the walls of an enclosure is

$$q_r = \frac{0.1723}{144 \times 3600} E_s' \left[E_g \left(\frac{T_g}{100}\right)^4 - a \left(\frac{T_s}{100}\right) \right] \qquad [9]$$

The emissivity and absorptivity of the gas mixture may be calculated from the values for the individual gas components, when the ratio of the absolute gas temperature to the absolute wall temperature is greater than 1.24 by the following equations:

$$E_g = (Eg_c \cdot C_c + Eg_w \cdot C_w - \Delta E_g) \qquad [10]$$

and

$$a = (a_c + a_w - \Delta E_s) \approx (E_{s_c} \cdot C_c + Eg_w \cdot C_w - \Delta E_s) \qquad [11]$$

The values of the gas emissivities and their correction factors were obtained, as mentioned earlier, from the extrapolation of the Hottel charts. The partial pressures of the water vapor and carbon dioxide were assumed to be those corresponding to the gas composition presented in (1). For a rocket motor, the length of the radiant beam through the gas mass is equal to the diameter of the combustion chamber.

Over-all Heat Transfer: The total rates of heat transfer q_T for the combustion chamber and for the nozzle throat were obtained by adding the rates of convective and radiant heat transfer. Thus,

$$q_T = q_c + q_r$$

Fig. 8 presents the calculated values

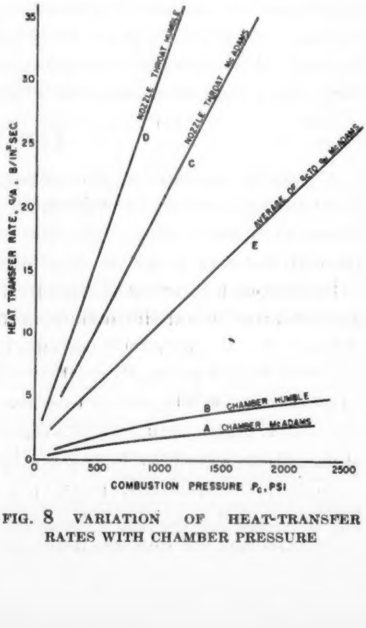

FIG. 8 VARIATION OF HEAT-TRANSFER RATES WITH CHAMBER PRESSURE

for the total rate of heat transfer as a function of combustion pressure. The curve A for the heat flux in the combustion chamber is in better agreement with the few experimental data than is the correlation based on Equation [7]. All of the curves indicate that while the heat flux in the combustion chamber increases with a rise in combustion pressure, the values do not become excessive for the pressure range studied, indicating that regenerative cooling should be adequate for the combustion chamber.

The curve C for the over-all heat-flux density in the nozzle throat section indicates that with high combustion pressures the heat-flux densities become so large that the effectiveness of regenerative cooling becomes doubtful. It is difficult, however, to measure the heat-flux density in the nozzle throat. Consequently, the data for heat fluxes for rocket-motor nozzles are usually presented as average values for the entire nozzle. In order to have some sort of basis for comparing the calculated values with the few available data, the authors calculated the curve E, the arithmetic mean flux density for the entire nozzle. The general shape of a rocket-motor nozzle is such that the heat flux for the entire nozzle has a value lying somewhere between the local value for the throat section and the local value for the entrance section. It was assumed that the heat-flux density in the entrance section does not differ significantly from that for the adjoining section of the combustion chamber.

The calculated curves of heat transfer to the rocket-motor combustion chamber and exhaust nozzle indicate that the heat fluxes in the nozzle throat become extremely large as the chamber pressure is increased. The upper limit of combustion pressure, where regenerative cooling of the nozzle becomes marginal, appears to be approximately 1200 psi for the motors studied. For operation at higher combustion pressures, it appears that some other form of cooling will be necessary.

Film Cooling

A possible solution to the cooling problem when the combustion temperatures are extremely high is to protect the metal walls of the rocket motor by a liquid film. The film can be established by injection of fluid through holes or a slot as illustrated in Fig. 9. The required length between slots is a function of the rate of evaporation of the liquid forming the film and the film thickness. Experiments, conducted by C. M. Beighley and W. L. Knuth at Purdue University, demonstrate that there is a maximum or critical amount of liquid that can be injected. If more liquid than the critical quantity is injected, the liquid film becomes unstable and breaks away, a phenomenon which is due probably to the

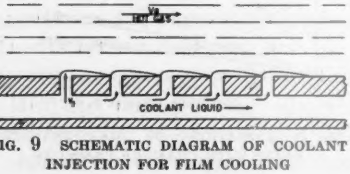

FIG. 9 SCHEMATIC DIAGRAM OF COOLANT INJECTION FOR FILM COOLING

turbulence in the gaseous boundary layer. To establish the conditions where the liquid film breaks away from the wall, experiments were conducted on the injection of liquid through different slots located in a flat surface over which a stream of cold air was passed. The velocity of the liquid in the slot when the film breaks away has been termed the critical coolant velocity v_f. Different slot widths and liquids have been investigated. The correlation of the data for slots perpendicular to the gas flow is presented in Fig. 10, where the coolant parameter ϕ is plotted as a function of the air velocity perpendicular to the slot.

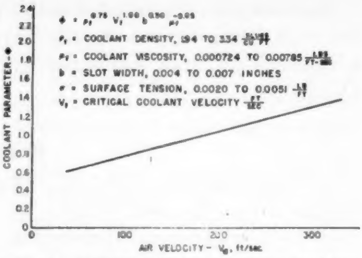

FIG. 10 ϕ VERSUS V_a FOR VARIOUS VALUES OF b, ρ_f, μ_f, AND σ

Conclusions

The results of the study indicate the following:

1 That the WFNA-(AN-F-58) propellants should give a performance comparable to that obtainable from the WFNA-aniline propellants.

2 There are no unusual problems to be solved in developing a reliable ignition system.

3 Regenerative cooling of the combustion chamber should be feasible even at combustion pressures of 2000 psia.

4 The cooling of the nozzle throat section by regenerative means will probably be inadequate at combustion pressures above 1200 psia.

5 The substitution of AN-F-58 for aniline as the fuel does not aggravate the cooling problem.

6 Considerable research, theoretical and experimental, is needed to establish the basic factors controlling heat transfer under conditions of large temperature differences between the hot fluid and the metal wall.

7 Data on the properties of gases at high temperatures are needed.

8 The effect of pressure on radiant heat transfer merits investigation.

9 The possibilities of film cooling appear attractive.

10 It is possible to establish a stable film flowing along a wall.

Acknowledgments

The authors express their indebtedness to G. M. Palmer, instructor, School of Aeronautics, Purdue University, and to Major R. W. Hoffmann U.S.A.F., graduate student, for their assistance in making many of the calculations upon which this paper is based.

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American Rocket Society News

Tentative Program for ARS 1950 National Convention

THE 1950 National Convention of the American Rocket Society will be held at the Hotel Statler, New York, N. Y., November 30—December 1, in conjunction with the annual meeting of The American Society of Mechanical Engineers.

The technical program will consist of three sessions devoted to theory, operations, and testing and design.

The tentative program follows:

THURSDAY, NOV. 30

Morning

Theory Session

A General Method for the Calculation of Theoretical Performance of Rocket Engines, by V. N. Hugg, National Advisory Committee for Aeronautics,

Lewis Flight Propulsion Laboratory, Cleveland, Ohio.

Stability of Liquid Films for Cooling Rocket Motors, by M. J. Zucrow, professor of gas turbines and jet propulsion, C. M. Beighley and E. L. Knuth, School of Mechanical Engineering, Purdue University, Lafayette, Ind.

Notes on the Behavior of Supersonic Gases in Overexpanded Nozzles, by K. Scheller and J. A. Bierlein, rocket development engineers, USAF, Air Materiel Command, Wright-Patterson Air Force Base, Dayton, Ohio.

Optimum Thrust Programming for a Sounding Rocket, by H. S. Tsien, Goddard professor, and Robert C. Evans, Guggenheim Jet Propulsion Center, California Institute of Technology, Pasadena, Calif.

Heat Recovery and Thermodynamic Efficiency in a Rocket, by engineer from, Battelle Memorial Institute, Columbus, Ohio.

Afternoon

Operations Session

Operation with the High Altitude Sounding Rocket *Viking*, by J. P. Layton, Glenn L. Martin Company, Baltimore, Md.

Rocket-Engine Flight Testing by R. F. Gompertz, aeronautical rocket propulsion engineer, Power Plant Branch, Edwards Air Force Base, Muroc, Calif.

Naval Air Rocket Test Station—Purpose and Progress, by Lt. Comdr. F. C. Durant, 123 USNR, engineering officer, U. S. Naval Air Rocket Test Station, Lake Denmark, Dover, N. J.

FRIDAY, DEC. 1

Morning

Testing and Design Session

Micro-Scale Rocket Studies, by Bradford Darling and Saul Wolf, Division of Industry Co-Operation, Massachusetts Institute of Technology, Cambridge, Mass.

Some Measurements of the Burning Rates of Mixed Liquid Bi-Propellants, by L. Greiner, H. Haussman, G. R. Wakepeace and C. W. Taitt, Naval Ordnance Test Station, Inyokern, Calif.

Design of Turbopumps for ATO Rockets (approximate title) by C. C. Ross, Chief Engineer, Liquid Engine Departments Aerojet Engineering Corporation, Azusa, Calif.

Throttling Thrust Chamber Control by M. Meyer, Supervising Project Engineer, Rocket Department, Curtiss-Wright Corporation, Propeller Division, Caldwell, N. J.

Membership Roster to Be Published

THE American Rocket Society is preparing a new roster of members which should soon be ready for distribution.

Each member of the Society has received a postcard form requesting the latest information on addresses and business connections. The secretary will appreciate prompt return of these cards. Every effort will be made to have the roster complete and accurate. Your co-operation will be appreciated.

ARS Junior Award

THE distinction of receiving an American Rocket Society Junior Award carries with it a recognition of meritorious achievement early in one's professional career. Papers are now being accepted at the New York Office for consideration for the 1950 Award. The award and medal will be presented at the Annual Convention to be held in December. To be considered, papers should be submitted not later than Oct. 31, 1950.

The winning paper will be judged mainly on the basis of content which should reflect original thought and effort. The age of authors of papers is limited to twenty-five. Papers should be on standard size paper, typewritten, and clearly marked: Submitted for Junior Award Competition. Send papers to: Secretary, American Rocket Society, 29 West 39th Street, New York 18, N. Y.